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DESIGN AND ANALYSIS OF A HIGHLY EFFICIENT POWER SYSTEM FOR THE SMALL SCIENTIFIC SATELLITE (S^3)

THOMAS A. LaVIGNA
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NOVEMBER 1969

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GODDARD SPACE FLIGHT CENTER
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ABSTRACT

In the design of a power system particular attention must be given to the interaction, both static and dynamic, of the various elements of the system. The method of combining the elements will not only greatly effect system efficiency but will also impose significant design constraints. Specifically, it has been shown that for maximum efficiency in a solar array-battery power system, the battery must be decoupled from the array bus so that excursions of battery voltage, during charge and discharge conditions, will not effect the array operating voltage and thus the available array power. In addition, the system efficiency can be increased by eliminating the main bus conversion step and using the array bus directly for loads.

With this criteria, a highly efficient solar array-battery power system for the Small Scientific Satellite (S^3) has been designed. Excellent utilization of array power is achieved by operating the array at a fixed voltage of 28 volts where the array is designed to provide maximum power. The control electronics for the system are simple, consisting of a shunt-charge regulator and a battery discharge regulator. Both function, as required, to maintain operation of the array at this fixed voltage and provide regulation of the main bus for all conditions of load. This system offers significant advantages of high efficiency, excellent array utilization, low electromagnetic interference, good static and dynamic load sharing and flexibility when compared to other known systems in use. It offers the power design engineer the opportunity to optimize the individual designs of the array, battery, and conversion electronics and tailor them to the particular mission.

Based on the S^3 power system developments and analysis which appears in this report, this system has been chosen for use on the Interplanetary Monitoring Platform (IMP) H, I, and J satellites and also the Planetary Explorer (PE) satellites.

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INTRODUCTION

The purpose of the Small Scientific Satellite (S^3) program is to develop a small versatile satellite that will accommodate a variety of experiments aimed at specific scientific investigations. Typical missions may require low altitude circular orbits or highly eccentric ones with apogees up to 30 Re.

With this in mind, efforts were made to develop for S^3 a highly efficient, flexible, and reliable power system capable of handling varying mission and load requirements. If sufficient consideration is given initially to these design aspects, then little change will be required for future missions. This would eliminate the need for costly design modifications.

Important in the design of a power system is a good understanding of how the elements of the power system (i.e., in this case, the solar array, battery, regulators, and converters) operate and how they interact with each other. Areas of importance that are considered are the behavior and power capability of the solar array as a function of load, and the effects of charge/discharge voltage characteristics of the battery on solar array performance. This study shows that several watts of power could be saved by decoupling the battery from the array by means of a series regulator. Further power savings can be achieved by the elimination of the main bus series regulator between the array bus and the main load bus. Selection of a primary bus voltage near that of the required load voltage further optimizes the efficiency of the system. Utilization of a high voltage system (such as 28 volts chosen) minimizes power losses in series diodes, transistors, etc.

POWER SYSTEM CONSIDERATIONS

This section deals with the important solar array battery and load interface considerations which effects system performance and efficiency. The three important interfaces which are considered are the array-battery, array-load, and the array-battery-load. For the array-battery interface, the effects of the battery charge profile on the array operating points must be evaluated. For the array-load interface, the requirements of the main power load must be considered in determining the array characteristics. For the array-battery-load interface, the operation of the system for loads in excess of the available array power must be analyzed to insure optimum system operation.

Matching the Battery to the Solar Array

Consider a solar array which has a typical I-V characteristic curve as shown in Figure 1. Over the normal operating range of the array (up to 27 volts), it behaves as a near constant current power supply. The selection and matching of the array and battery voltages are chosen to maintain the upper charge voltage of the battery, and hence, the maximum operating voltage of the array, just to the left of the maximum power point at position A. The region to the right of the maximum power point is not used because the array power decreases sharply with a slight increase of voltage.

An 18 cell silver cadmium battery has a charge profile as shown in Figure 2. It is characterized by a two step voltage level charging profile. These voltage levels are known as the silver-monoxide and the silver-peroxide levels. The battery at the silver-monoxide level behaves as a near constant current load at about 1.2 volts per cell over a large charge current and temperature range. However, once the second level is achieved (about 35% charge), the voltage level becomes very dependent on charge current and temperature, and must be voltage limited.

If the battery is shunted directly across the solar array and if it is in the lower silver-oxide state, then it will control the voltage of the array as shown by point B (21.5 volts) in Figure 1 and will determine the maximum amount of power to be delivered by the array. In this example, it would be 21.5 watts as compared to the maximum power capability of 27 watts at the upper battery operating voltage of 27 volts.

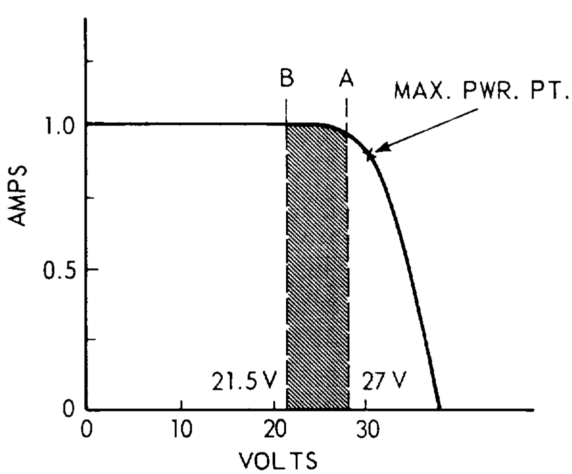

Figure 1. A Typical Solar Array I-V Characteristic

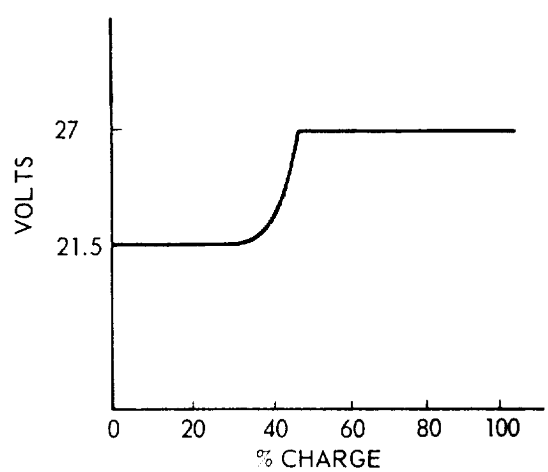

Figure 2. A Typical Charge Profile for an 18 Cell Silver Cadmium Battery

In a normal mode of operation, the battery will be in the silver mon-oxide state when the spacecraft exits a shadow, and the array will be required to supply sufficient power to sustain loads and charge the battery. For example, if the spacecraft load is 18 watts and the power required to charge the battery is 3 watts, then the array must supply at least 21 watts at 21.5 volts (battery silver monoxide charging voltage state) and would have the capability of delivering 27 watts (1 amp × 27 volts) at the upper operating voltage. Hence 5.5 watts (27.0 w - 21.5 w) is not available from the array under this charging condition and thus a 27 watt array is required to support a 21.5 watt spacecraft load because of array-battery voltage mismatch. However, if the battery was decoupled from the array by means of a series regulator so that it did not load the array; and the array bus was maintained at 27 volts under all load conditions, then a significantly smaller array could be chosen and would provide 21 watts at 27 volts. This represents a savings of 5.5 watts or 20% in the size of the array. With an estimated solar array power to weight ratio of 2.2 watts per pound then a weight saving of 2.5 lbs is realized.

Matching the Solar Array to the Load

An important aspect which must be considered in the selection of the regulated bus voltage is the requirements of the largest power consuming subsystem. However, in applying this consideration, care must be taken to prevent the load requirements, such as ripple, regulation, etc., from seriously affecting and thus constraining the design of the system. If the solar array bus can be used directly to supply this power, a series power conversion step will be eliminated and, thus, a significant improvement in efficiency can be realized. A further advantage, resulting from the elimination of an added power conversion stage, is that of increased reliability.

Another factor which has a significant effect on system efficiency is the array-main bus interface, specifically, the range of voltage over which the array will operate. For reasons of reliability and isolation, blocking diodes are used to isolate the array sections from each other. These diodes, introduce voltage drops and associated power losses. Since this power loss is directly proportional to the array current, it can be kept to a minimum by operating the array at low current (high voltage). However, certain array reliability considerations — such as limiting the number of cells in series — will restrict the upper limit of voltage. Thus, a tradeoff review was required with primary consideration given to the efficiency of power transfer from array to the bus.

Array Battery Load Interface

In this section, the interaction of the array, battery, and load are analyzed. Specifically, the effects of applying a load which exceeds the array capability are considered. For such a situation, it is important that the array deliver as much power as possible; thus, it is important that the operating voltage of the array be maintained higher than the discharge voltage of the battery. For illustrative purposes of explaining the interaction of the battery and the array an 18 cell AgCd battery is considered with maximum charge voltage of 27 volts (upper plateau), minimum charge voltage of 24 volts (lower plateau), and a discharge voltage cut off of 16.2 volts.

Figure 3 shows the effects of the battery on the array for two conditions of load. For an 18 watt load the array is capable of charging the battery and satisfying the load requirement. Considering a 32 watt load as shown in Figure 3, it can be seen that while the array is capable of 27 watts at 27 volts, it cannot deliver 27 watts to the load since the battery under a slight load will decrease to 24 volts (or less) and therefore clamp the array at 24 volts (or less). For such a condition, the array will supply 24 watts and the battery will supply the additional 8 watts. If the array was maintained at 27 volts, at which it supplied 27 watts, then the battery would need to supply only 5 watts.

A similar analysis for a near depleted battery condition (Figure 4) shows that, under worst conditions, the array voltage can be decreased to 16.2 volts (undervoltage condition) with only 16.2 watts available and the battery would have to supply 15.8 watts. These conditions demonstrate the advantages of isolating the battery from the array.

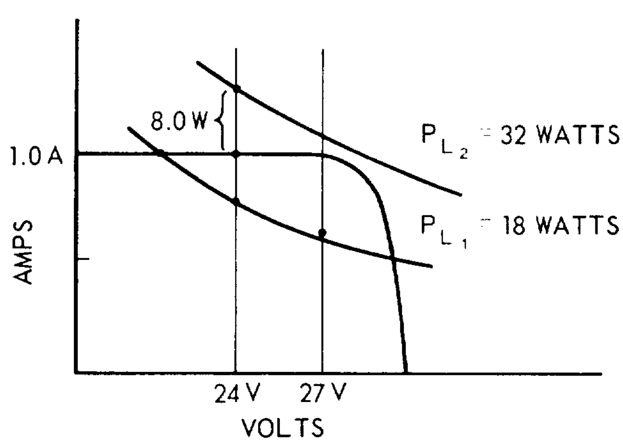

Figure 3. Influence of the Battery on the Array Operating Points for 2 Load Conditions

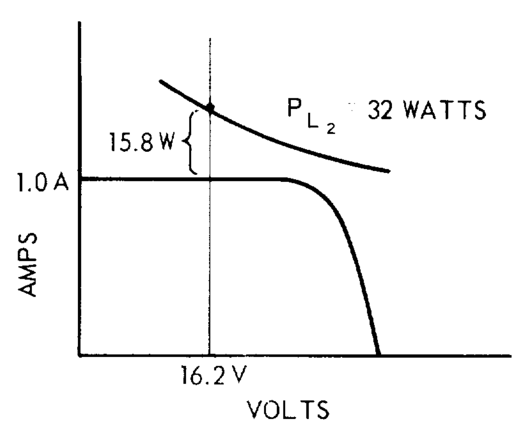

Figure 4. Influence of A Depleted Battery and Heavy Load on the Array Operating Point

POWER SYSTEM REQUIREMENTS

Mission Requirements

The orbit for the first mission, S^3-A, will be elliptical with a perigee of about 150 n. miles, an apogee of about 5 earth radii (geocentric), an inclination of 3 degrees and an orbital period of 7-1/2 hours. Nominal shadow periods will be about 33 minutes in duration although some periods of up to 1-1/2 hours may occur. There also may be seasons of continuous sunlight.

For other missions, typical orbits could be: (1) of the low altitude circular variety (several hundred kilometers) with a period of about 90 minutes and inclination anywhere from equatorial to polar, or (2) highly eccentric with apogee up to 30 Re, a period up to 4 days and an inclination from 0° to 40° or more. Shadow periods could range from 40% per orbit to 0% per orbit for the low altitude orbits. For the eccentric orbits, shadow periods occurring at perigee could range up to about 3/4 hour although seasonal effects could produce weeks of 100% sunlight as well as extended apogee shadows of up to 8 hours duration, on several consecutive orbits.

Spacecraft Configuration

The basic physical configuration of the S^3 spacecraft, in its simplest form, is a polyhedral shaped structure (see Figure 5) which approximates a 27 inch diameter sphere. It has 26 flat surfaces. All but a few of these surfaces will be made available for the placement of solar panels. This configuration provides a nearly uniform power output over the sun/spin angle ranges of 0 to 180 degrees. However, if an attitude control system is employed, (which is an option), and sun/spin angle constrained to a given region of space, then it might be advantageous to alter the configuration of the spacecraft or make use of flip-out panels. For example, if the mission requirements imposed a sun/spin angle range 90° ± 20°, then it would be advisable to employ a configuration which resembles a right cylinder. This can be accomplished by providing cylinderical skirts about the top and bottom section. On the other hand, if mission requirements dictate a sun/spin angle of 0 ± 20 degrees, it would be wise to maximize the solar array power by flipping out the bottom panels to an optimum angle. Depending upon selection, the above configuration will enable the solar array to provide power in the 20 to 40 watt range. Additional power output above this range could be achieved by supplementing the body mounted array with solar paddles.

S^3-A will employ a magnetic orientation control system to control the sun/spin angle within 20 to 70 degrees. The primary reason for this restriction is

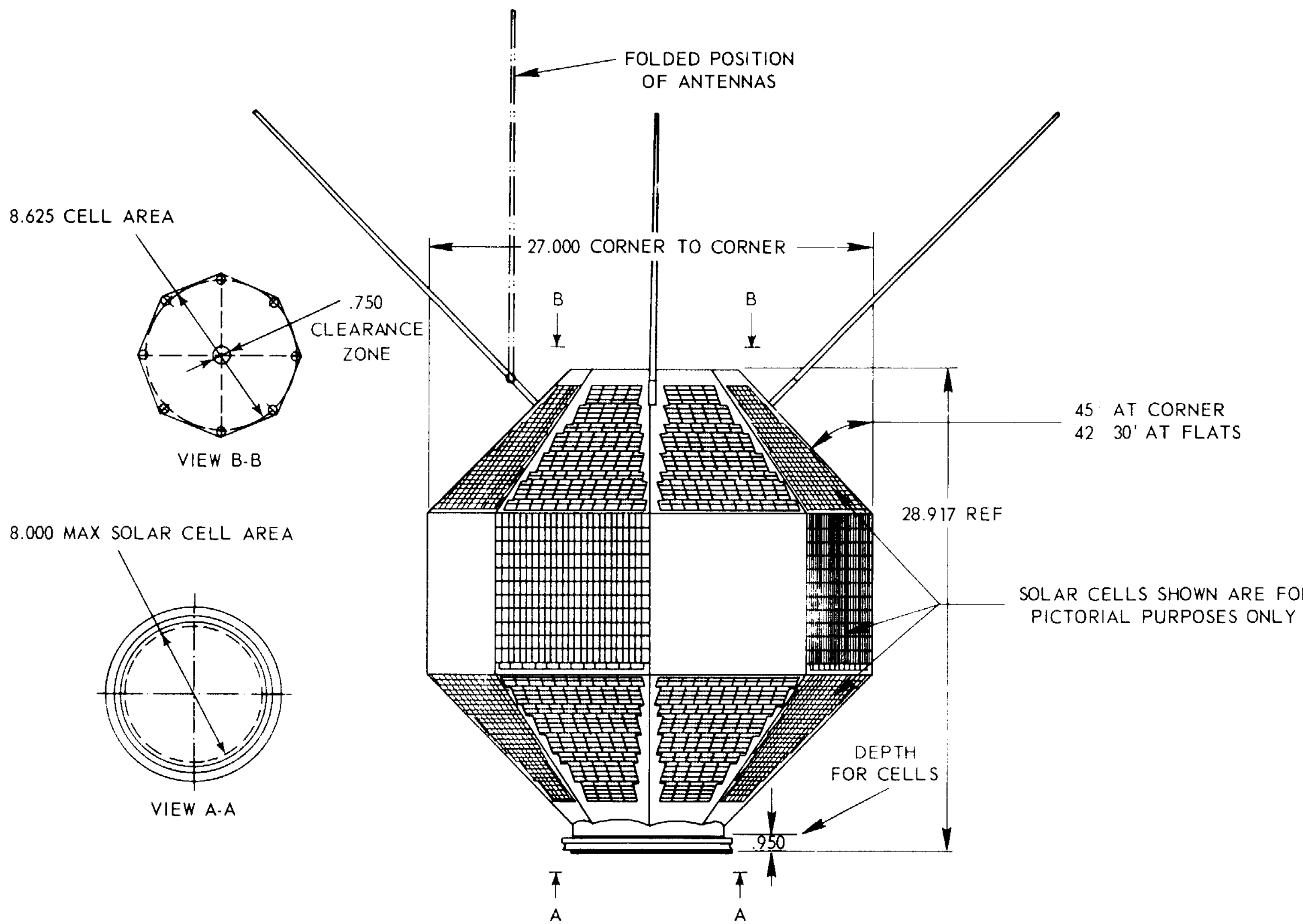

Figure 5. Basic S^3 Physical Configuration

the use of particle detectors which look out along the equator and spin axis of the spacecraft with long angles of 0 ± 20 degrees. These regions are forbidden to sunlight due to the possibility of severe damage to the detectors. The solar array configuration for S^3-A has been optimized for the 20 to 70 degree range of spin-axis-sun angle.

The conditioning and control units of the power subsystem will be housed in the octagonal midsection of the spacecraft. They will be packaged in trapezoidal frames that are plugged into the electrical harness in the central area of the octagon. The battery will be located in the center of the spacecraft and ample volume is provided for expansion to larger capacity batteries for future missions.

Electrical Requirements

The first S^3 mission will require 18 watts of continuous power during the tape recording mode, (approximately 97% of orbit time), and 31 watts during the playback mode (approximately 3% of orbit time).

The power budget breakdown for S^3-A is shown in Table 1. To minimize battery loading, the tape playback mode of the spacecraft will be constrained to the sunlight portion of the orbit with only occasional playback modes occurring in shadow. A power profile for a typical orbit is shown in Figure 6.

The following are the electrical requirements of the power system:

a. Load Power Output — The output load power shall range from 5 to 32 watts (potential increases to 60 watts for follow-on missions).

b. Voltage Input — The input voltage will be defined from array/battery interface necessary to satisfy the load requirements at an output bus voltage of 28 volts.

c. Operation — The limits to the following characteristics shall pertain under any or all combinations of the normal operating conditions (see a, b) and applicable environmental requirements:

   (1) Static Regulation — The output voltage shall remain within a $\pm 2\%$ band of the nominal 28 volt output.

   (2) Dynamic Regulation — The dynamic regulation requirements of the main bus are shown in Table 2.

d. Output Ripple — The output ripple of the main bus shall be kept at a minimum. The body of the ripple shall not exceed 50 millivolts peak to

Table 1

S^3 Power Budget

Item	Record Mode	Playback Mode
Loads Operating From Inst. Conv.	Watts	Watts
1. Data Handling System	4.6	4.6
2. Data Sync Clock	0.77	0.77
3. Transmitter	0.5	0.5
4. Receiver and Command Decoder	0.8	0.8
5. Cmd. and Power Programmer	0.45	0.45
6. Optical Aspect	0.69	0.69
Sub Total	7.81	7.81
7. Inst. Conv. Losses (85% Eff.)	1.38	1.38
	9.19	9.19
Loads Operating From 28 Volt Main Bus		
8. Transmitter	1.2	13.0
9. Tape Recorder	1.3	2.4
10. Experiments	6.2	6.2
	8.7	21.6
Total	17.89 w	30.79
	≈18 w	≈31 w

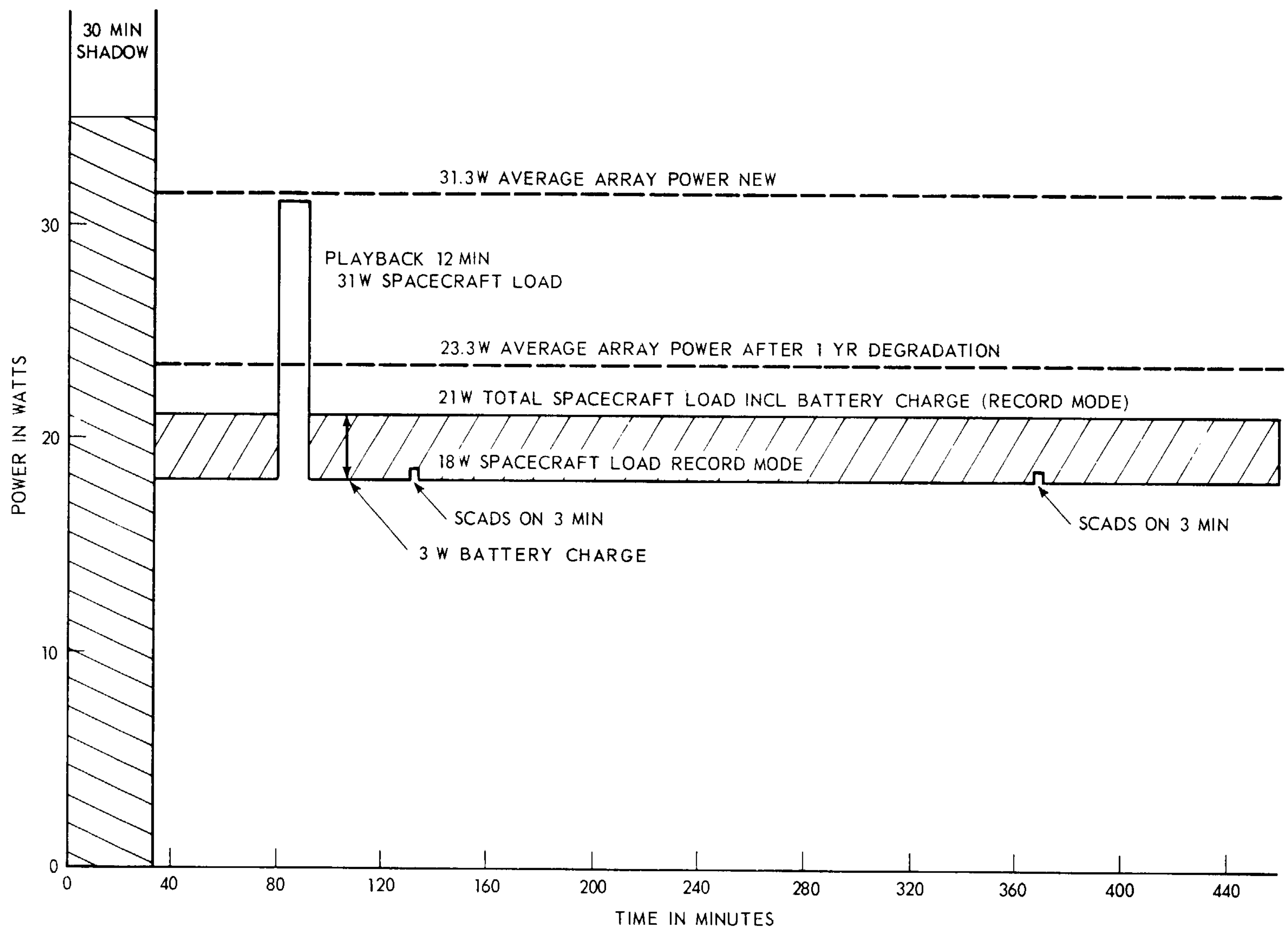

Figure 6. Normal Load Operational Modes and Power Profiles Per Orbit

Table 2

Dynamic Regulation

Condition	Maximum Voltage Overshoot	Maximum Voltage Undershoot	Maximum Response Time
(1) Input voltage: operating range of solar array and battery step wave with $t_{rise} = t_{fall} < 1.0$ ms.	10%	10%	0.2 ms
(2) Load current: 0.2 to 2.0 amperes step wave with $t_{rise} = t_{fall} < 50 \mu s$.	10%	10%	0.5 ms

peak, while the total ripple including spikes shall not exceed 75 millivolts peak to peak at the worst operating condition. For a load condition within the 5 to 0 watt band, the output ripple including spikes shall be limited to 200 millivolts peak to peak.

e. Input Ripple Current — The ripple current reflected back onto the array or battery bus shall be limited to 20 milliamps peak to peak into a one ohm source impedance.

f. System Frequencies — All regulators or converters in the primary power system shall operate at 20 KHertz ±5% for any or all combinations of the normal operating conditions. However, provisions shall be included to accommodate future missions which require ±2% regulation.

SELECTED POWER SYSTEM DESIGN

Basic Power System

The basic power system of S^3 consists of a solar array, battery, solar array shunt regulator, battery charger, and battery discharge regulator. A block diagram of the power system is shown in Figure 7.

The solar array, which primarily supplies power for the spacecraft loads, must also provide power to charge the spacecraft battery. The major portion of

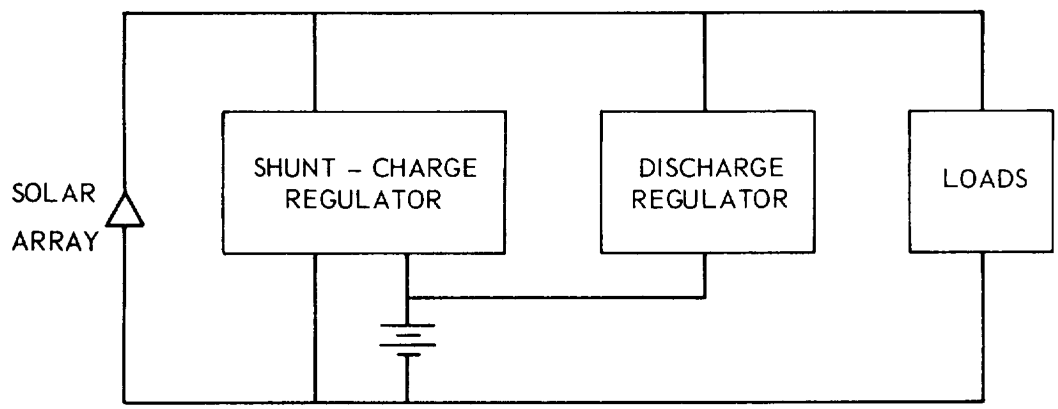

Figure 7. Basic S^3 Power System

the array power is passed directly to the loads, while a smaller portion appropriately controlled by the shunt-charge regulator is provided for charging the battery. A discharge regulator is used to regulate the bus voltage when battery power is needed.

The shunt portion of the regulator is used to limit the maximum bus voltage to 28 volts +2% by dissipating the excess solar power not required by the spacecraft loads or battery. The charge control portion functions to regulate battery charge in accordance with a prescribed charge method for the type of battery used. In order to prevent overloading of the array causing the bus voltage to drop below 28 +0.5% during battery charging, the regulator is provided with bus voltage feedback which effectively limits the current to that available from the array at 28 volts. The battery and the discharge regulator are used to limit the bus voltage to a minimum of 28 volts -2% during periods when the spacecraft loads exceed the capability of the array. When the available array power is adequate, the regulator will remain in a standby operating condition sensing the main bus voltage.

Figure 8 shows the regulation bands in which the charge and discharge regulator each function. A separation band of 0.28 volts is maintained between the operating regions of the shunt charge and discharge regulator. This region is required to prevent both the shunt charge regulator and discharge regulator from being active and both attempting to regulate the bus voltage.

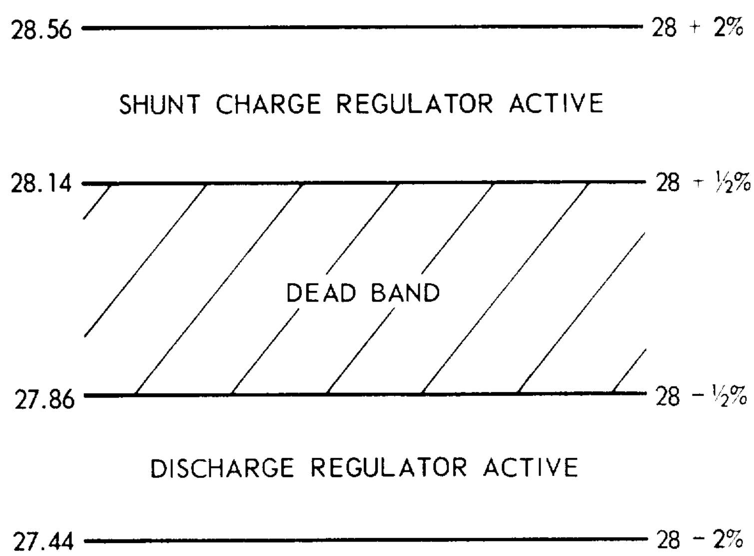

Figure 8. Regulation Bands for the S^3 Power System

System Performance and Characteristics

Figure 9 shows a typical solar array curve. Superimposed upon the array curve is the input characteristic of the combination shunt-charge regulator. As explained previously, the operating voltage of the array is clamped to 28 volts +2% by the shunt-charge regulator and is limited to 28 volts -2% by the battery and discharge regulator.

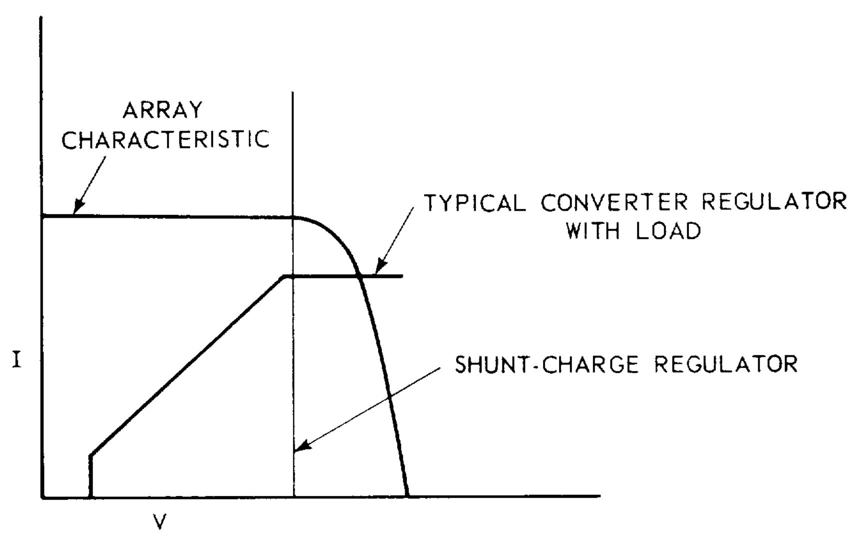

Figure 9. Graphical Representation of the System Characteristics

Most loads used on S^3 are interfaced to the main bus through individual power supplies consisting of a dc to dc converter with series regulators. The input characteristic imposed by this interface is shown in Figure 9.

The operating point on the array is fixed at a nominal 28 volts and the array current available at this voltage is supplied to the load. Any excess is used to charge the battery or is dissipated. This operating point along with the load current and charge-shunt current is shown in Figure 10 for a condition in which the array power exceeds the load demand. Figure 11 shows the case where the array power is less than the load requirements. In this case the charge-shunt current is zero. The array operating point is limited by the discharge regulator to 28 volts -2% and the deficient power is supplied by the battery through the discharge regulator.

Elements of the Power System

Battery. The battery for S^3-A will be a series string of 18 rechargeable 3 A.H. silver-oxide-cadmium cells. This defines a battery bus with an upper charge voltage of 27 volts and a minimum discharge voltage of 16.2 volts. However, for other S^3 missions, the power system is designed to accept nickel-cadmium batteries as well. For an 18 cell Ni-Cd battery this range would be 18.0 to 26.0 volts. The number of cells in series will be standardized at 18 for all missions resulting in standard battery bus voltages. The selection of 18 cells in series was made because it places the upper charge voltage of the battery within 1 volt of the main bus (28 volts) and thus minimizes the power losses in the charge regulator. Also power losses in the discharge regulator are less when the number of cells in series are maximized. The ampere hour capacity of the battery may differ for various missions and is dependent on such mission parameters as peak power demand, eclipse power demand, and degree of solar cell shadowing by booms. Using the criteria that the depth of discharge should not exceed 25% to insure 1 year lifetime, the A.H. capacity required can be calculated. For the S^3-A mission the anticipated load requirement is 12 watt hours per orbit. An 18 cell 3 A.H. Ag-Cd battery will yield 57 WH. when subjected to the S^3-A mission requirements. The depth of discharge (12/57) is calculated to be 21%.

A 2-level voltage regulation scheme will be used for charge control of the battery. Using this scheme, the battery voltage during charge will be limited to 27.0 volts until the battery charge current decreases to* c/100 (40 ma ± 10 ma for the 3 A.H. Batt), at which time, the battery voltage will be reduced to 25.2 volts (see Figure 12).

*c = ampere-hr. capacity

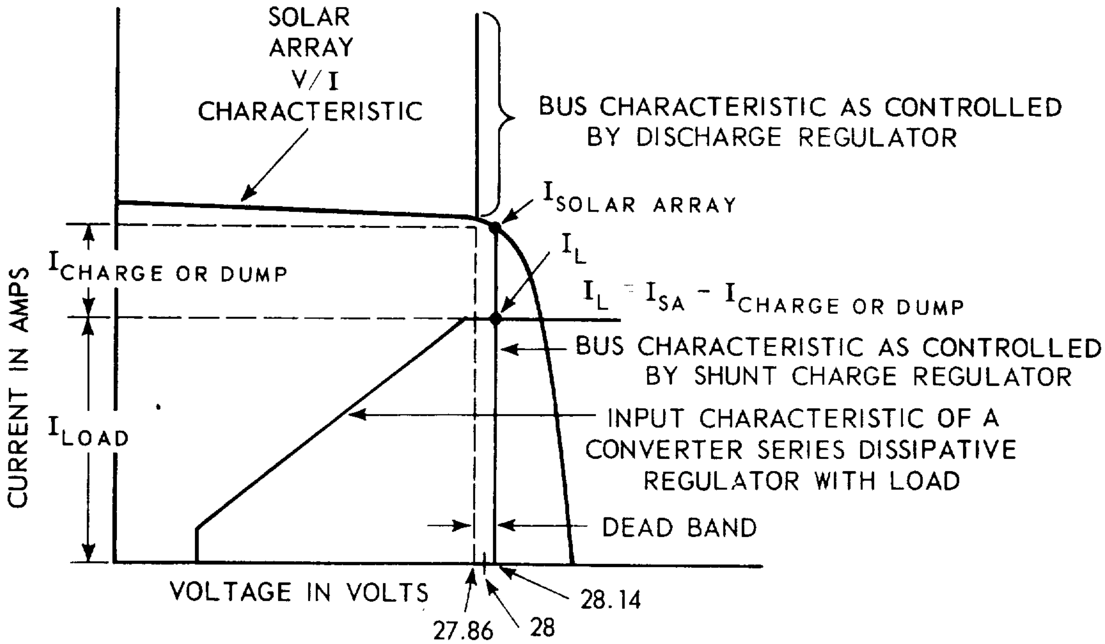

Figure 10. Graphical Representation of the System Characteristics for a Condition in Which the Array Power Exceeds the Load Power

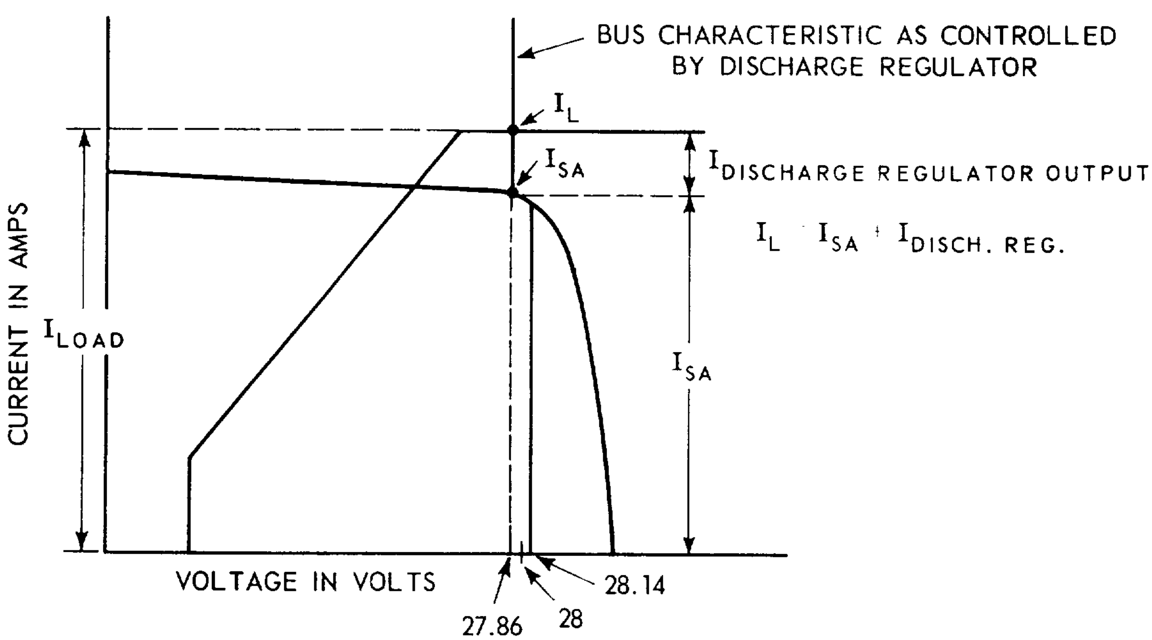

Figure 11. Graphical Representation of the System Characteristics for a Condition in Which the Array Power Is Less Than the Load Power

The silver-cadmium battery is characterized by two distinctive voltage levels defined as the silver-monoxide (Ag_2O) and silver-peroxide (AgO) plateaus. The silver monoxide level is present until 30 to 40% charge is reached, at which time, the battery quickly changes to the silver-oxide level. While the battery is in the Ag_2O condition, it behaves as a constant potential charging load, i.e., over the temperature range of 0° to 25°C and charging current of c/20 to c/3, the voltage changes only by 0.9 volts from 21.5 to 22.4 volts. However, its voltage is very sensitive to temperature and current and therefore, the battery voltage must be clamped to avoid overcharging. Prolonged charging at the clamped voltage of 27 volts (the suggested clamp voltage to properly charge the battery at 1.5v/cell) will induce cell voltage unbalance, with resultant pressure build up. To avoid this problem the battery voltage is reduced to 25.2 volts (open circuit voltage of the battery) when the charging current decreases to c/100 (about 95% charge).

The discharge characteristics of the battery are shown in Figure 13. The battery will sustain the following load voltage plateaus:

Voltage	Temperature	Current
19.5V	25°C	1.0A
18.7V	25°C	2.0A
18.9V	0°C	1.0A
17.5V	0°C	2.0A

However, depending upon the previous history of the battery, (such as amount of time on charge, time since last usage) the voltage at the beginning of discharge can be lower than the average plateau voltage by 1.7 volts or higher by 3.0 volt as shown in Figure 12.

Another important characteristic of the battery is the shifting of the transition region between the Ag_2O and AgO charge levels (see Figure 14). A depleted battery, during charge, will exhibit this transition region at 35% A.H. capacity input. However, if slightly more than 10% of the capacity of a fully charged battery is removed, then the transition region is shifted to 85% to 90% capacity input. The S^3-A battery will be subjected to a 21% depth of discharge and therefore will exhibit this condition every orbit. This condition is mentioned because it influences the design of the power system. An initial look at the charge characteristic of the silver-cadmium battery with the transition region occurring at 35% capacity input would suggest that the battery would not load the array (if

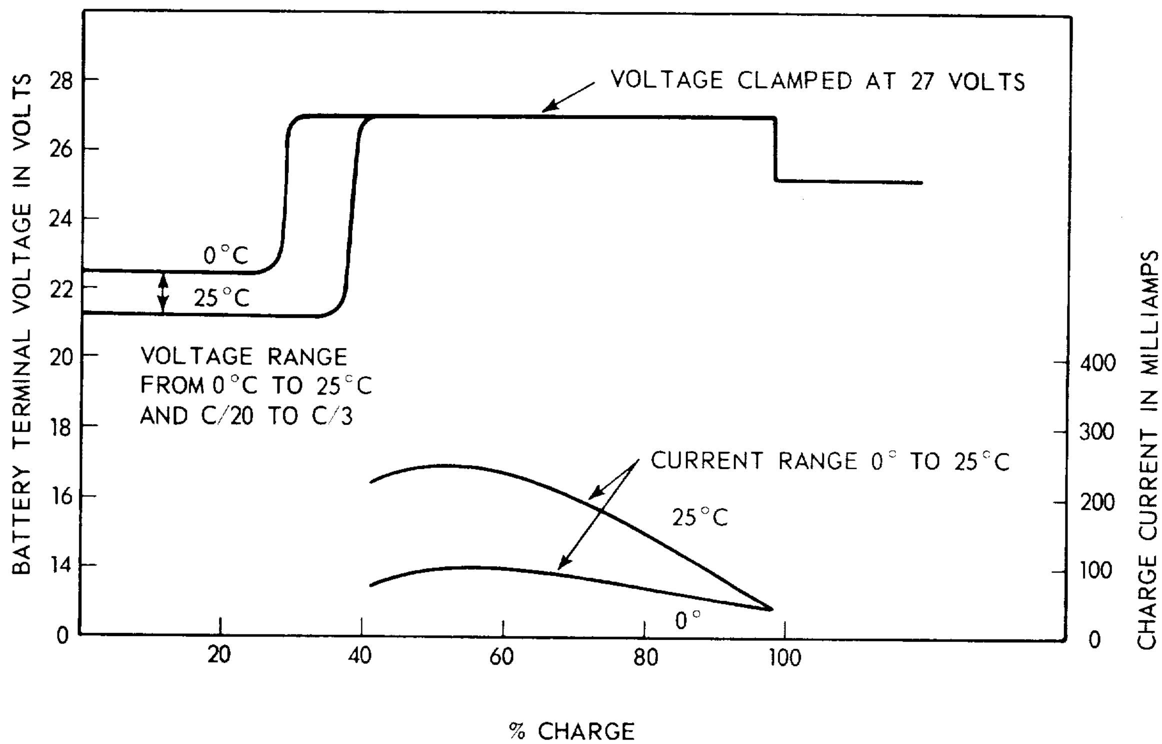

Figure 12. An 18 Cell Sil-Cad Battery Characteristic as a Function of Percent Charge

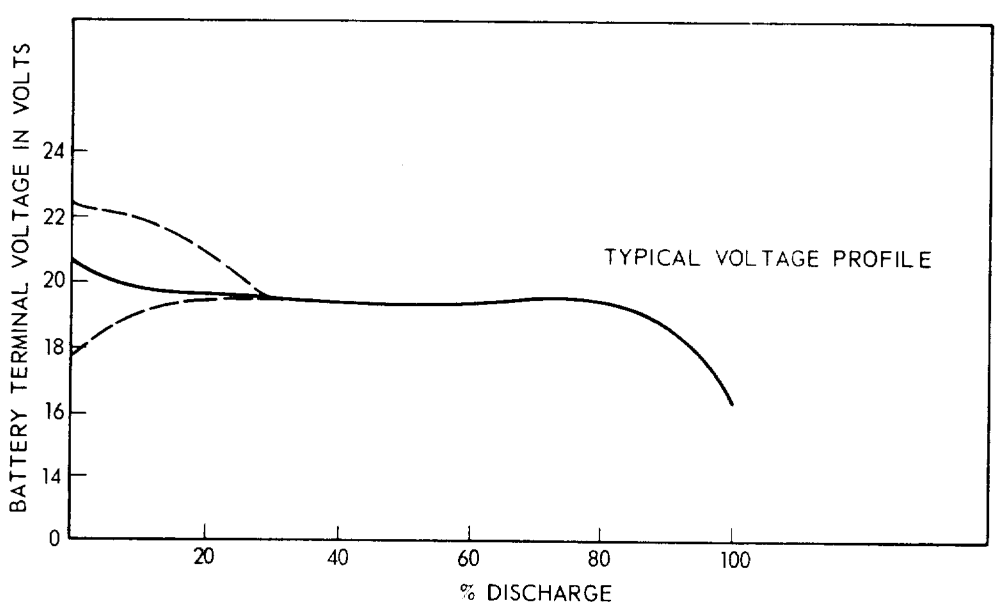

Figure 13. An 18 Cell Sil-Cad Battery Characteristic as a Function of Percent Discharge

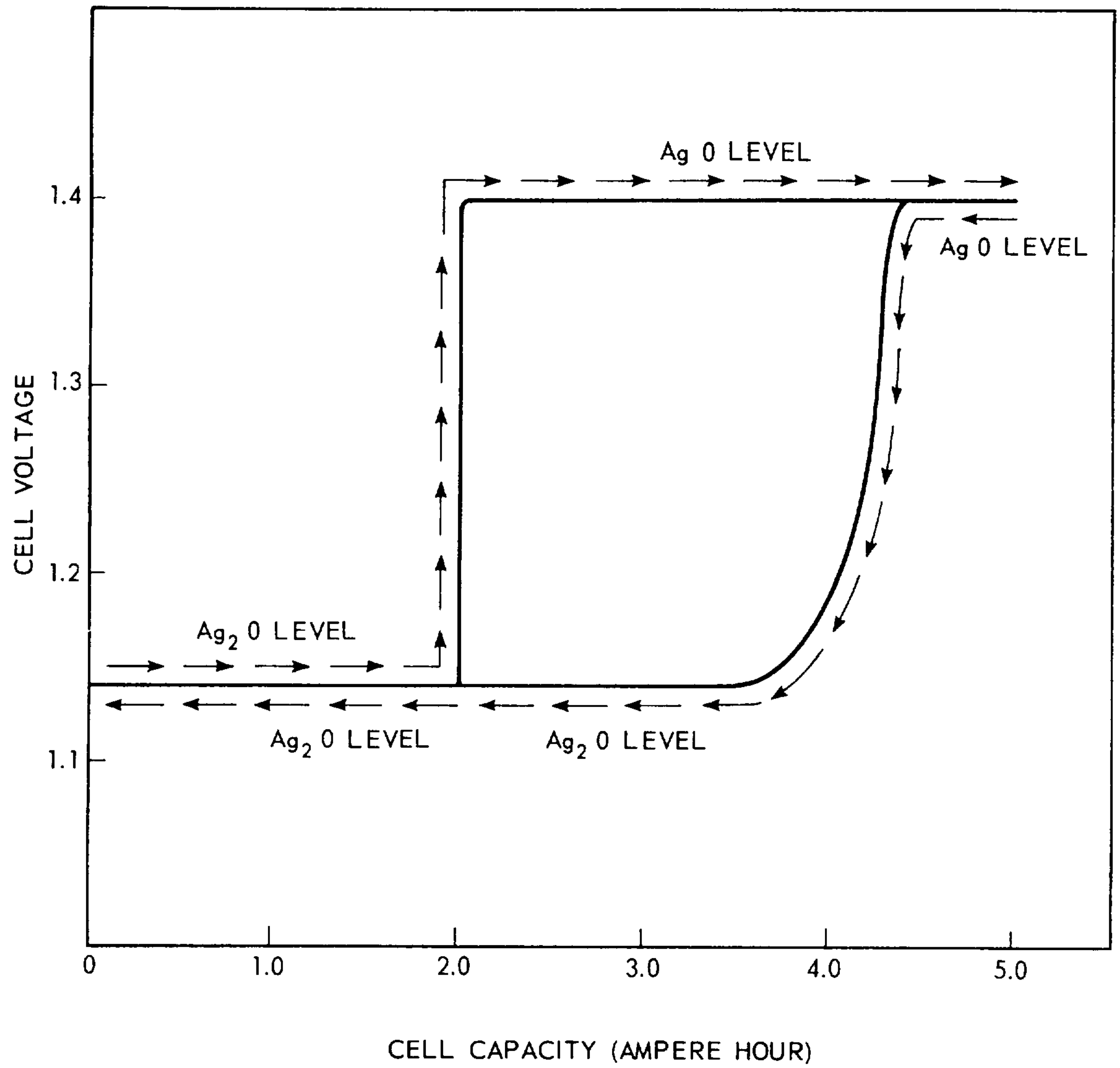

Figure 14. Open Circuit Battery Cell Voltage as a Function of Cell Capacity

connected directly to it) until at least 65% of the capacity was removed. Further reasoning would suggest that if the battery was subjected to a 20% Depth Of Discharge, then it would always remain at the AgO level while being charged. This, however, is not the case and the battery would load the solar array off the maximum power point every orbit. This condition again stresses the need for decoupling the battery from the array bus.

Battery Charge Regulator. The solar array shunt regulator and the battery charge regulator will be incorporated into one unit* and will function to regulate the battery charge power while also limiting the maximum solar array bus voltage. The reason for this incorporation is to insure proper voltage control on the 28 volt bus and minimize interference of the two in controlling the bus. To limit the bus voltage, a series dissipative regulator with a resistive load will be used. For battery charge control, the design will make provisions for use of either a switching regulator or a series dissipative regulator. The choice of regulator will depend on the type of battery employed (NiCd or AgCd) and the orbit. A series dissipative regulator, with its inherent simplicity, can be considered for the battery charge control because of its comparable efficiency to a nondissipative regulator. Calculations indicating this are shown below:

Charge Efficiency Calculations

(1) Dissipative regulator

(a) Efficiency during the silver monoxide charging condition:

$V_{in} = 28$ volts   Power available = 3 watts

$V_{battery} = 21.5$ volts $I_{available} = \frac{3}{28} = 0.107$ amps

$P_{loss\ for\ control} = 40$ mw

$P_{loss} = (0.107)(6.5) + 0.040 = 0.736$ watts

$P_{charge} = 3-0.736 = 2.264$ watts

$$\text{Efficiency} = \frac{2.264 \text{ watts}}{3 \text{ watts}} = 75.5\%$$

(b) Efficiency for charge during the silver oxide charging condition:

$V_{in} = 28$ volts   Power available = 3 watts

$V_{battery} = 27$ volts $I_{available} = 0.107$ amps

$P_{loss\ for\ control} = 35$ mw

$P_{loss} = (0.107)\ (1) + 0.035 = 0.142$ watts

*Suggested by J. Paulkovich, NASA, GSFC.

$$P_{charge} = 3 - 0.142 = 2.858 \text{ watts}$$

$$\text{Efficiency} = \frac{2.858 \text{ watts}}{3 \text{ watts}} = 95\%$$

(c) Total charge efficiency

$$0.35 \times 75.5\% = 26.4\%$$

$$0.65 \times 95\% = \underline{61.7\%}$$

$$88.1\%$$

(2) Nondissipative switching regulator

Overall Efficiency = 90%

(3) Comparison

90% Nondissipative Regulator

<u>88.1%</u> Dissipative regulator
1.9% ≈ 2% - advantage of nondissipative regulator

A 2 level regulator will be used for charge control of the battery. The voltage level on charge will be limited to 27 volts until the battery current decreases to 40 ma at which time the regulator will open and allow the battery to reach its open circuit voltage which will be approximately 25.2 volts.

A common steering circuit will be used to provide active control to the shunt-transistor-resistor network and the battery series transistor. This circuit will receive signals of the battery voltage, current, and bus voltage and will function to control the battery charge and shunt load simultaneously, with the battery charge as the primary control parameter. Of course, the bus voltage feedback will be the overriding control which will prevent loading of the array during battery charging. Figure 15 shows a basic diagram of the shunt-charge regulator.*

<u>Discharge Regulator</u>. The discharge regulator must process the battery power with its varying input voltage characteristic and provide a constant output voltage for all changing load conditions. The regulator will function to maintain the main bus voltage at 28 volts whenever the battery is required to supply power to the loads. This situation will occur when the load power demand

*Proposed by J. Paulkovich, NASA, GSFC.

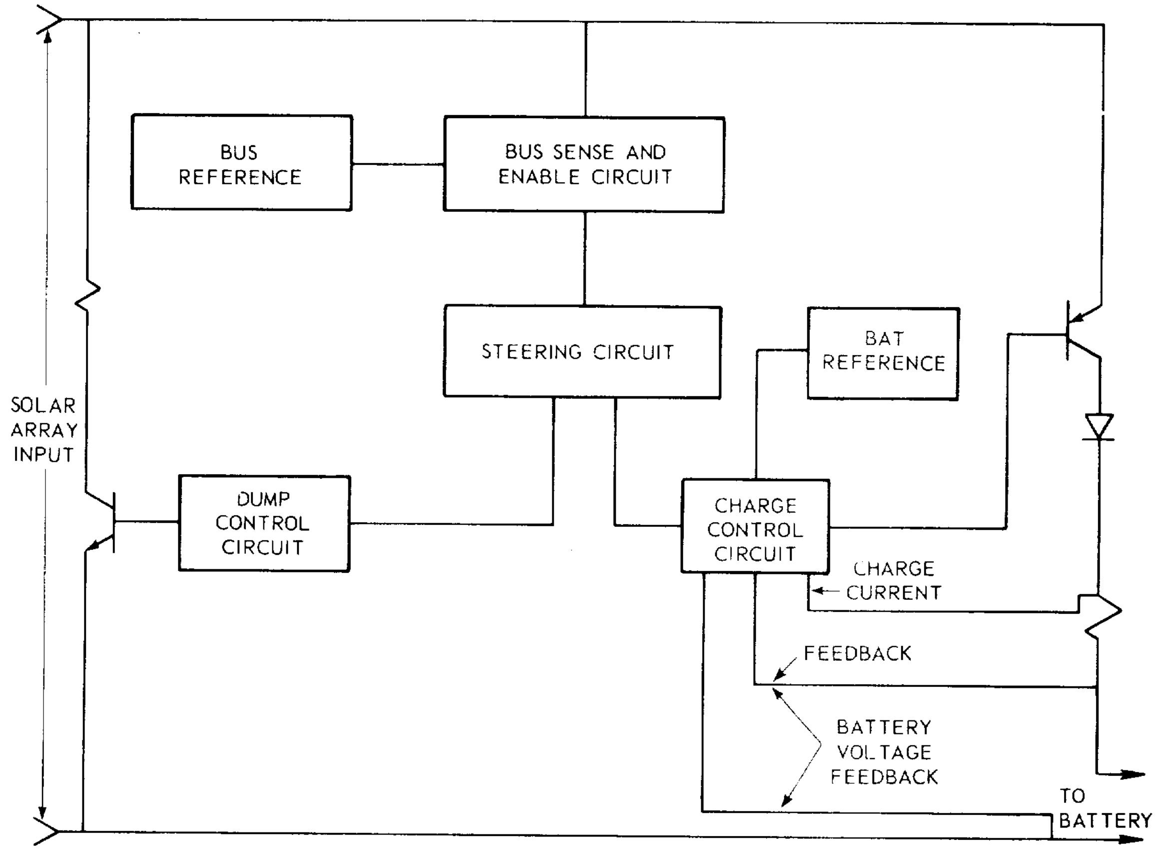

Figure 15. Basic Block Diagram of the S^3 Shunt-Charge Regulator

exceeds that power available from the array and during solar eclipse. During times when the available array power is adequate for supplying the loads, the regulator will remain in a standby operating condition sensing the main bus voltage. A decrease in the main bus voltage to a predetermined level will result in the instantaneous discharge of the battery through the battery boost regulator to satisfy the load deficiency. The level at which this regulated battery power will be supplied is selected to be well within the -2% lower limit of regulation.

A voltage boost regulator is used to accomplish regulation of the battery power since the bus voltage must be maintained at 28 volts ± 2%, which in all cases exceeds the battery voltage. The regulator is designed to handle continuously up to 60 watts of load power to allow for power growth and flexibility.

The basic regulator employs a power inverter paralleled with the battery to provide a voltage boost. This inverter is required to handle and switch only the boost power and not the entire load power, thus offering increased efficiency as compared to other boost power techniques. This inverter uses an auto transformer in series with the main input. With such a circuit configuration, the inverter need only provide a voltage boost, which when added to the input voltage, will result in the required average output voltage. Control of the inverter is provided by a pulse width modulated drive circuit which operates at a constant frequency of 20kHertz.

The regulator will be designed to provide ± 0.75% regulation for any and all combinations of normal operating conditions. Typical conversion efficiency for the anticipated load and line variation is estimated to be greater than 90%. Figure 16 shows a basic diagram of the discharge regulator.

Solar Array. The solar array is composed of 14 rectangular panels and 8 trapezoidal panels, all mounted in 3 bands around the spacecraft. The 8 trapezoidal panels form the top band. 6 rectangular panels and 2 equal and opposite

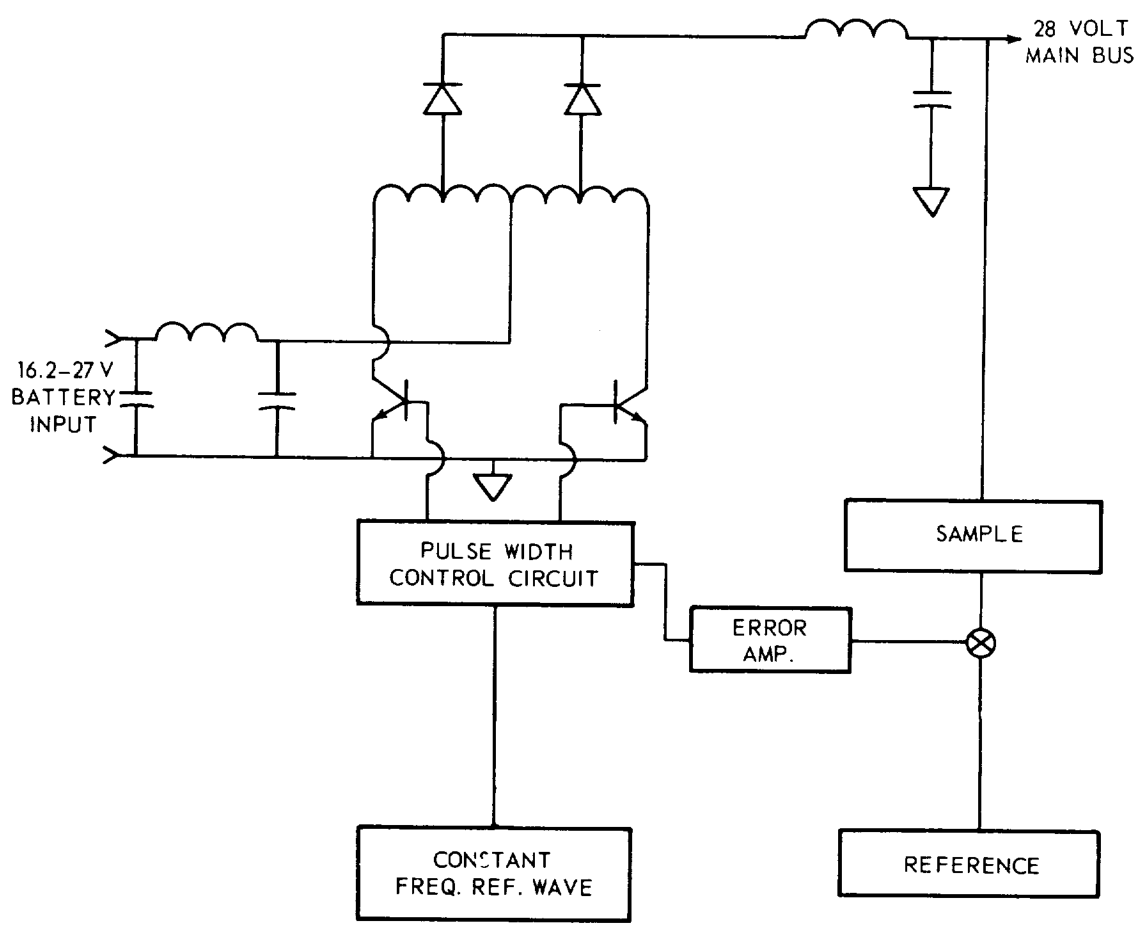

Figure 16. Basic Block Diagram of the S^3 Discharge Regulator

blank spaces comprise the middle band, and the remaining 8 rectangular panels form the bottom band. Each rectangular panel is perpendicular to the equatorial plane of the satellite while each trapezoidal panel is at a 47.5 degree angle to it.

In the array, 800 nominal 2 by 2 cm and 1116 nominal 2 by 3 cm N-P solar cells are used. Each trapezoidal panel consists of 100 of the 2 by 2 cm cells connected in series. The rectangular panels in the middle band consist of 82 of the 2 by 3 cm cells connected in series. The bottom band panels consist of 78 of the 2 by 3 cm cells also connected in series. Each solar cell has a protective coverslide 40mils thick to reduce radiation damage.

All solar panels are connected in parallel and are electrically isolated from each other by blocking diodes. Each solar cell is shunted by a bypass diode to minimize the power loss due to the shadowing effects of the appendages and also to increase array reliability.

The solar cells to be used have a minimum air mass zero conversion efficiency of 10 percent at 30°C. The solar array has an estimated average output of 24.5 watts at sun axis/spin (S/S) angle of 20 degrees, increasing to a 31.1 watt average at 70 degrees. The radiation environment is expected to reduce the power output of the array at the end of 1 year to an 18.4 watt average at an S/S angle of 20 degrees and 23.3 watts average at an S/S angle of 70 degrees.

Since the mission requires that the spin axis-sun angle be constrained between 20 and 70 degrees, the original spacecraft configuration (a polyhedral structure that approximates a 27 inch diameter sphere with 26 flat surfaces) was modified by adding skirt panels on the lower band of the spacecraft to boost array output over this region. Figure 17 shows the S^3-A spacecraft with skirt panels.

System Flexibility

In following the basic S^3 philosophy of system flexibility (that is, the ability of the system to accommodate varying mission requirements) certain provisions in the power system were made. The regulators for battery charge control and conversion electronics are designed to handle power increases to 60 watts. These increases will require increased watt hour battery capacity by the use of larger ampere hour capacity cells while maintaining the number of series cells fixed at 18. The system can also accommodate Ni-Cd cells (with coulometer or third electrode sensing by minor modification to the battery charge regulator. Increased power from the array can be achieved by supplementing the body mounted array with fold out panels or paddles.

For low orbit missions (90 minute orbit) which use NiCad batteries and array paddles, which are subjected to very low temperatures (to -60°C) during

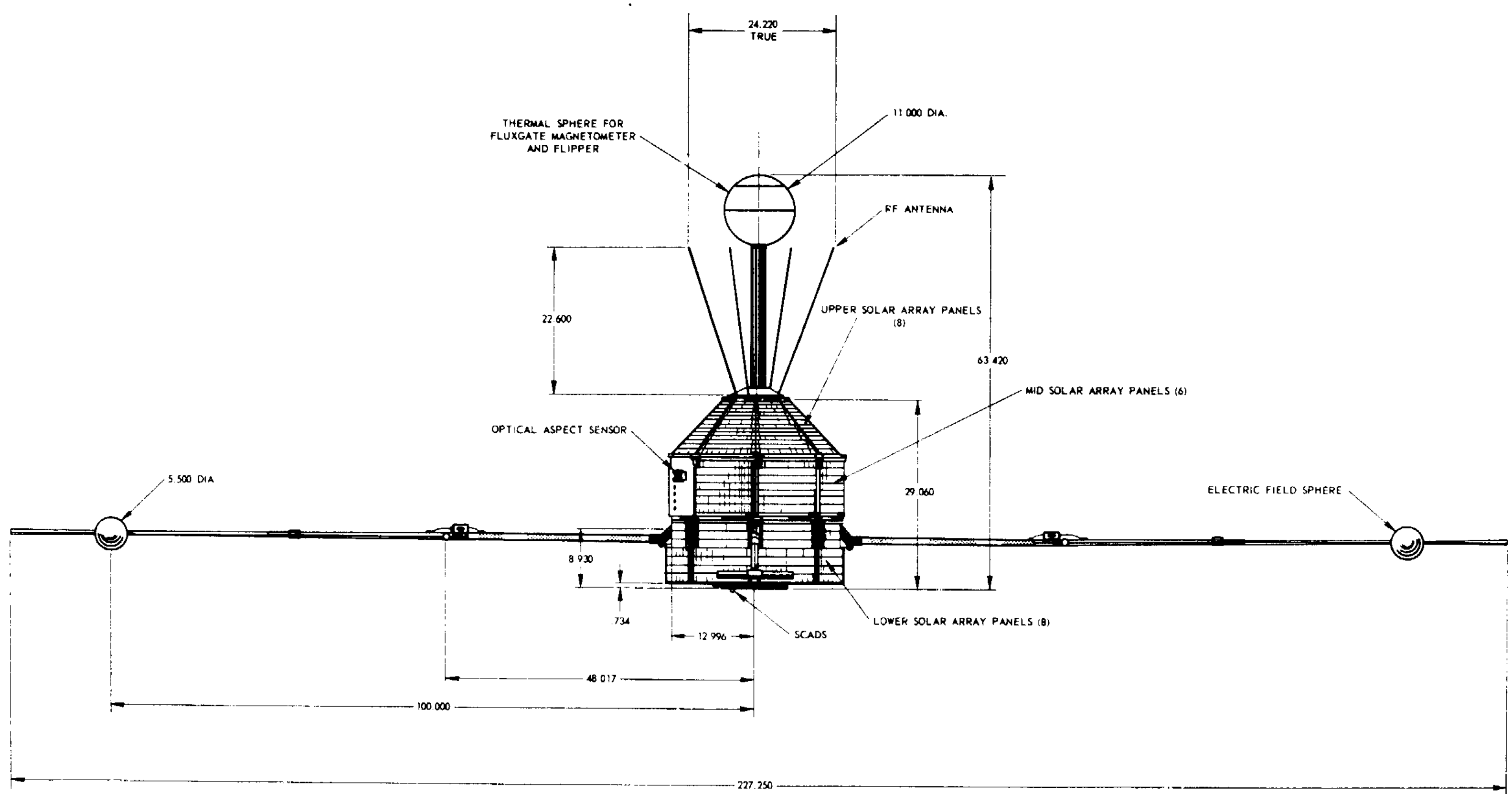

Figure 17. S^3-A Spacecraft Physical Configuration

shadow, the system can be modified to take advantage of the additional power available from the cold array at the times of exit from shadow. For a paddle mounted array which reaches temperatures of -60°C in shadow, a 20% increase in average power can be realized in a 90 minute orbit. However, for a body mounted array as on S^3-A where the low temperatures are limited by good thermal transfer to the spacecraft structure, only a 10% increase in average power can be realized. Figure 18 shows this system configuration.

This system will use a series switching regulator between the array and the battery bus to control the operating point on the array characteristic as a function of battery charge and array capability. This regulator operates similar to the shunt charge regulator by employing input voltage feedback to prevent overloading of the array. Feedback signals from a coulometer or 3rd electrode normally control the switching regulator duty cycle and, thus, the operating point on the array characteristic when the array power available exceeds or equals the load and charge requirements at or above a preselected array voltage. At this preselected array voltage, a feedback input signal will override the battery charge sensing signals and control the output power. Thus, by clamping the array voltage under maximum load demand the regulator will function to deliver, up to and for many conditions above, 90% of the available array power.

A voltage boost regulator identical in all respects to the discharge regulator, described in the chosen system, is used to couple the load to the optimum array power regulator and battery. Thus the entire load requiring regulation interfaces to the power system through this regulator.

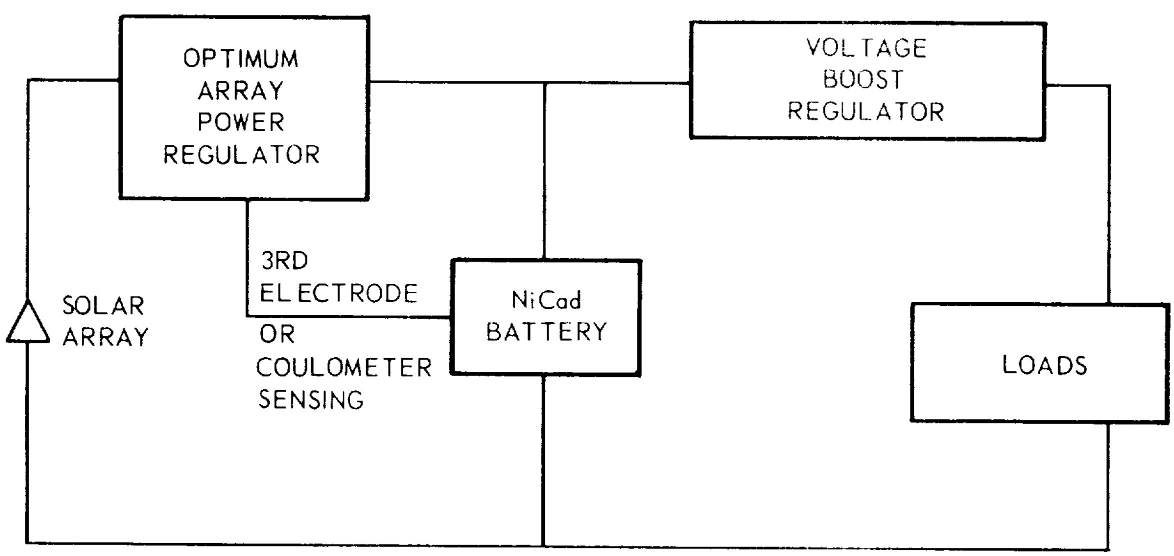

Figure 18. A Modified Version of the S^3 Power System For Use With Ni-Cad Batteries in Low Orbits

The system, while employing two series switching regulators between the array and loads, will provide operation very near to the array peak power and thus afford maximum utilization of the available array power.

EFFICIENCY COMPARISON OF THE SELECTED POWER SYSTEM TO OTHERS IN USE

This section of the paper is devoted to developing the proposed S^3-A power system efficiency and comparing it to the efficiency of other systems in use such as the shunt battery-array-regulator system (IMP, RAE) and the maximum array power tracking system. In addition, the effect on efficiency of future mission requirements, such as lower orbits, shall be evaluated. The efficiencies of the 3 systems considered are calculated for 3 possible S^3 orbital missions — 7.5 hour orbit (S^3-A), 90 minute orbit, and 4 day orbit.

I. Selected Power System

The proposed S^3 power system as described in this paper is repeated in Figure 19.

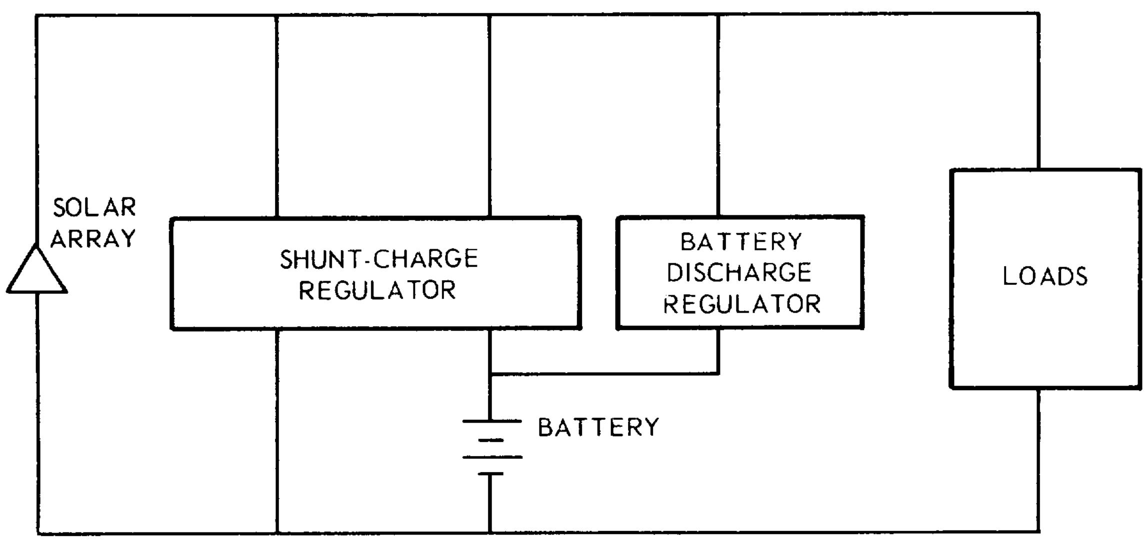

Figure 19. Basic S^3 Power System

A. <u>S^3 System 7.5 Hour Orbit</u>

<u>Condition A</u>: Shadow (30 minutes)

During this condition, the only elements of the main power system that are active are the battery and the battery discharge regulator (BDR). The system operates in both the record and playback modes.

		Record Mode	Playback Mode
(1)	S/C Loads	18 watts for 18 min.	31.00 Watts for 12 min.
(2)	Losses Disch. Reg. (90%)	2 watts	3.45 Watts
(3)	Power required from battery	20 watts	34.45 Watts

(4) Energy required from battery

$$20 \text{ w.} \times \frac{18}{60} \text{ hr.} = 6.0 \text{ WH}$$

$$34.45 \text{ w.} \times \frac{12}{60} \text{ hr.} = 6.9 \text{ WH}$$

Total 12.9 WH

<u>Condition B</u>: Sunlight (7 hours)

Normally in this mode the active elements of the power system will be the solar array, shunt regulator and battery charger. The spacecraft will be considered to be in the record mode.

(1) Energy required to charge battery

$$\frac{12.9 \text{ WH}}{75\% \text{ batt. eff.} \times 90\% \text{ chrg. eff.}} = 19.2 \text{ WH}$$

(2) Power required to charge battery $\frac{19.2 \text{ WH}}{7.0 \text{ H}} = 2.75$ watts

(3) Spacecraft loads 18.00 watts

Total array power req. 20.75 watts ≈ 21 watts

System Efficiency Calculation

To provide a means for comparing the systems considered for this application, two efficiency calculations can be made. The first, which is the conversion efficiency gives the efficiency by which power is processed from the power sources. It is calculated from the efficiency of the main bus regulator stages, the battery charge and discharge regulators, and the battery under charge and discharge conditions to satisfy the load requirements. The second, which is the system efficiency, is a measure of the array power required to satisfy the load and provide battery charge. It is a function of the conversion efficiency, time duration of sun and shadow periods per orbit, and the battery charge requirements.

$$\text{Conversion Efficiency} = \frac{(E_{LR})\,(t_L)}{t_T} + \frac{(E_{BC})\,(E_{BD})\,(E_B)\,t_D}{t_T}$$

$$\text{Conv. Efficiency} = \frac{(100)\,(7.0)}{7.5} + \frac{(90)\,(90)\,(75)\,(0.5)}{7.5} = \underline{97.5\%}$$

$$\text{System Efficiency} = \frac{\text{Average load power per orbit}}{\text{Array power required}}$$

$$\text{Average load power per orbit} = P_1 \frac{t_1}{t_T} + P_2 \frac{t_2}{t_T}$$

$$\text{Average load power per orbit} = 18\left(\frac{7.3}{7.5}\right) + 31\left(\frac{0.2}{7.5}\right) = 18.376 \text{ w}$$

$$\text{System Efficiency} = \frac{18.376 \text{ watts}}{21 \text{ watts}} = \boxed{87.6\%}$$

where:

E_{LR} - load regulator efficiency

E_{BC} - battery charge regulator efficiency

E_{BD} - battery discharge regulator efficiency

E_B - battery efficiency

t_L - duration of sun time

t_D - duration of shadow time

t_T - total orbit time

t_1 - duration of Record Mode

t_2 - duration of Playback Mode

P_1 - Spacecraft load in Record Mode

P_2 - Spacecraft load in Playback Mode

B. <u>S^3 System 90 Min. Orbit</u>

<u>Condition A</u>: Shadow (30 min.)

		Record Mode		Playback Mode
(1)	S/C Loads	18 watts for 24 min.		31 watts for 6 min.
(2)	Losses BDR	2 watts		3.45 watts
(3)	Power required from battery	20 watts		34.45 watts
(4)	Energy required from battery	$20 \text{ w.} \times \frac{24}{60} \text{ hr.}$	=	8.00 WH
		$34.5 \text{ w.} \times \frac{6}{60} \text{ hr.}$	=	3.45 WH
				11.45 WH

<u>Condition B</u>: Sunlight (60 minutes)

(1) Energy required to charge batt. $= \frac{11.45 \text{ WH}}{0.75 \times 0.90} = 17 \text{ WH}$

(2) Power to charge battery $\frac{17 \text{ WH}}{1 \text{ H}} = 17$ watts

(3) S/C loads 18 watts

Total array power req. 35 watts

System Efficiency Calculation

$$\text{Conv. Efficiency} = \frac{(100)\ (60)}{(90)} + \frac{(90)\ (90)\ (75)\ (30)}{(90)} = \underline{86.8\%}$$

$$\text{Average Load} = \frac{(18)\ (84)}{90} + \frac{(31)\ (6)}{90} = 18.87 \text{ watts}$$

$$\text{System Efficiency} = \frac{18.87 \text{ watts}}{35 \text{ watts}} = \boxed{54\%}$$

C. <u>S^3 System 4 Day Orbit (32 Earth Radii)</u>

<u>Condition A</u>: Shadow (30 minutes)

Energy required from battery = 12.9 watt hours

<u>Condition B</u>: Sunlight (95.5 hours)

(1) Energy to charge battery $= \frac{12.9 \text{ WH}}{0.75 \times 0.90} = 19.2$ watt hrs.

(2) Power to charge battery $= \frac{19.2 \text{ WH}}{95.5 \text{ H}} = 0.20$ watts

(3) Spacecraft loads 18.00 watts

Total array power req. 18.20 watts

System Efficiency Calculation

$$\text{Conv. Eff.} = \frac{(100)\ (3.995)}{4} + \frac{(90)\ (90)\ (75)\ (0.005)}{4} = \underline{99.9\%}$$

$$\text{System Efficiency} = \frac{18 \text{ w}}{18.20 \text{ w}} = \boxed{98.9\%}$$

II. <u>Shunt Battery Array Power System (IMP Power System)</u>

The IMP spacecraft A to G employed a shunt type system as shown in Figure 20.

This system consists of a solar array, shunt regulator, 13 cell battery, and prime converter. The prime converter operates at an average efficiency of 85%. The source or primary side of the converter operates over the voltage range of 12 to 19.6 volts as defined by the charge and discharge characteristics of the battery.

A. <u>IMP System 7.5 Hour Orbit</u>

<u>Condition A</u>: Shadow (30 minutes)

During this condition the battery discharges at an average voltage of 14 volts through the prime converter. The system operates in both the record and playback modes.

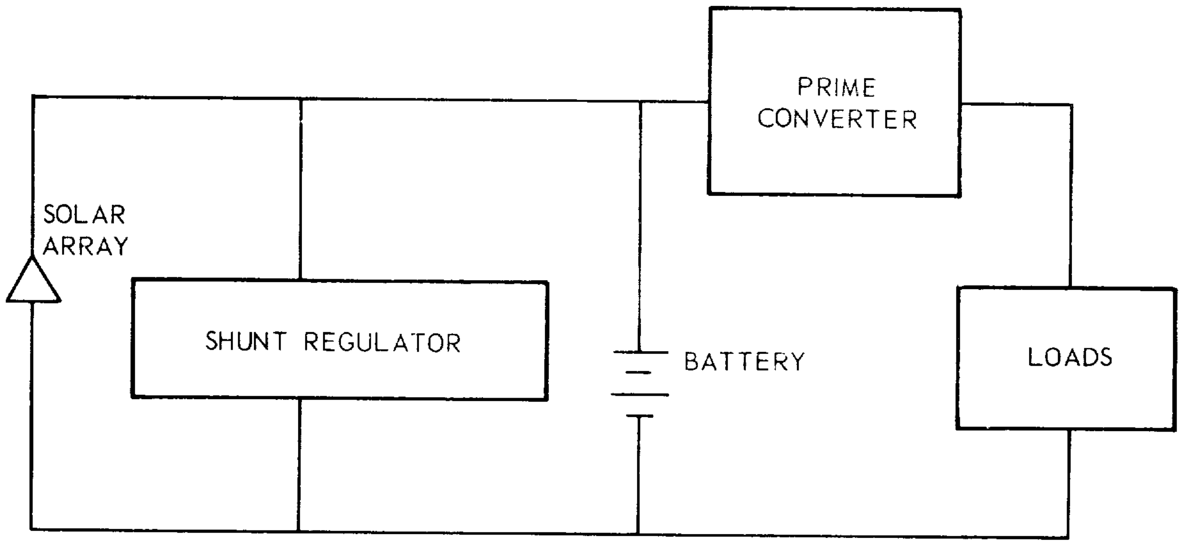

Figure 20. Basic Shunt Array Battery Power System Used on RAE and IMP

		Record Mode	Playback Mode
(1)	Spacecraft loads	18 watts for 18 min.	31 watts for 12 min.
(2)	Losses P.C. (85% eff)	3.2 watts	5.50 watts
(3)	Power required from battery	21.2 watts	36.50 watts
(4)	Energy required from battery	$21.2 \text{ w.} \times \frac{18}{60} \text{ hr.} =$	6.37 WH
		$36.5 \text{ w.} \times \frac{12}{60} \text{ hr.} =$	7.30 WH
			13.67 WH

Condition B: Sunlight (7 hours)

The spacecraft will normally be in the record mode.

(1) Energy required to charge battery $\frac{13.67 \text{ WH}}{75\% \text{ batt. eff.}} = 18.25 \text{ WH}$

(2) Power required to charge battery $\frac{18.25 \text{ WH}}{7.0 \text{ H}}$ = 2.61 watts

(3) Spacecraft loads including prime conv. losses = 21.2 watts

23.81 ≈ 24.0 watts

(4) Excess power required due to array-battery interface (from Figure 21) 6.3 watts

Total array power req. 30.3 watts

When the spacecraft exits from shadow, the array is required to supply sufficient power to sustain loads and charge the battery. The array must deliver this power at 15.5 volts, the battery charge voltage in the silver monoxide level. The calculations that follow show that the array will have an excess of 6.3 watts @ 19.6 volts.

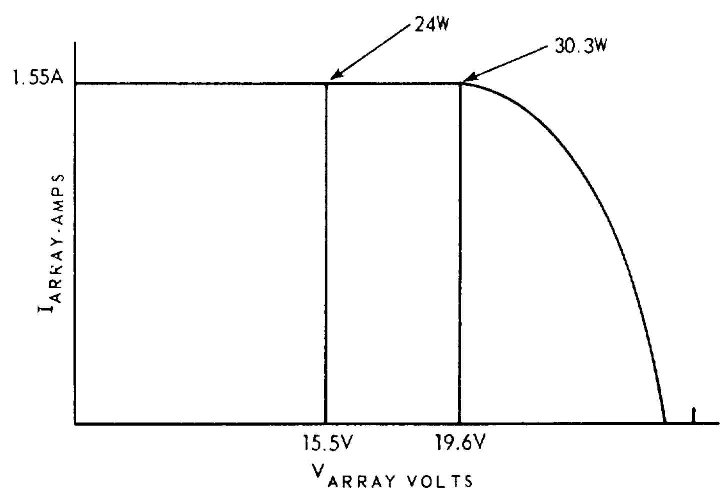

Figure 21. Battery-Array Loading Effects for the IMP Power System in a 7.5 Hour Orbit

Array power at 15.5 volts = 24 watts

$$I_{array} = \frac{24 \text{ W}}{15.5\text{v}} = 1.55\text{A}$$

P_{array} @ 19.6 volts = (1.55A) (19.6) = 30.3 watts

30.3 watts

-24 watts

6.3 watts — excess power required due to battery-array loading

System Efficiency Calculation

$$\text{Conv. Efficiency} = \frac{(85)\ (7.0)}{7.5} + \frac{(85)\ (75)\ (0.5)}{7.5} = \underline{83.4\%}$$

$$\text{System Efficiency} = \frac{18.376 \text{ watts}}{30.3 \text{ watts}} = \boxed{60.6\%}$$

B. <u>IMP System 90 Minute Orbit</u>

<u>Condition A</u>: Shadow (30 minutes)

		Record Mode		Playback Mode
(1)	S/C loads	18 watts for 24 min.		31 watts for 6 min.
(2)	Losses	3.2 watts		5.5 watts
(3)	Power required from battery	21.2 watts		36.5 watts
(4)	Energy required from battery	$21.2 \times \frac{24}{60}$	=	8.5 Watt Hours
		$36.5 \times \frac{6}{60}$	=	3.65 Watt Hours
				12.15 Watt Hours

<u>Condition B</u>: Sunlight (60 minutes)

(1)	Energy to charge battery	$\frac{12.15 \text{ WH}}{75\%}$	=	16.2 Watt Hours
(2)	Power to charge battery =	$\frac{16.2 \text{ WH}}{1 \text{ Hr.}}$	=	16.2 Watts
(3)	S/C loads including prime conv. losses			21.2 Watts
				37.4 Watts
	Excess power required due to array-battery interface (from Figure 22)		=	9.9 Watts
	Total array power req.		=	47.3 Watts

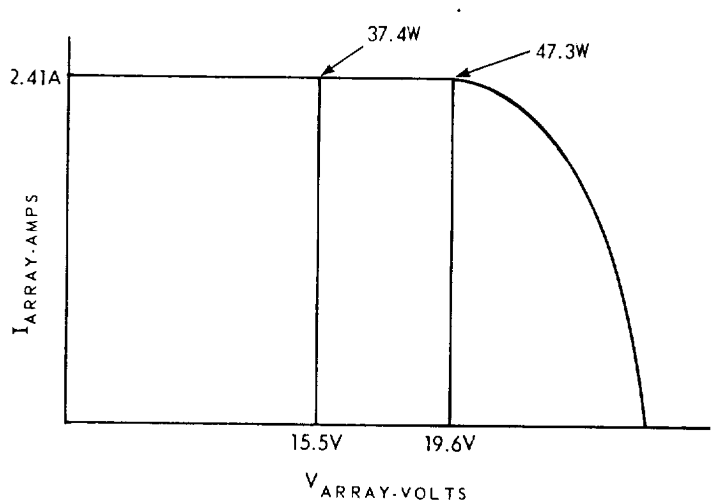

Figure 22. Battery-Array Loading Effects for the IMP Power System in a 90 Minute Orbit

System Efficiency Calculation

$$\text{Conv. Efficiency} = \frac{(85)\ (60)}{90} + \frac{(85)\ (75)\ (30)}{90} = 77.9\%$$

$$\text{System Efficiency} = \frac{18.87 \text{ Watts}}{47.3 \text{ Watts}} = \boxed{39.9\%}$$

C. IMP System 4 Day Orbit (32 Earth Radii)

Condition A: Shadow (30 minutes)

Energy required from battery = 14.54 Watt hours

Condition B: Sunlight (95.5 hours)

(1) Energy to charge battery $= \dfrac{14.54 \text{ WH}}{0.75} =$ 19.4 Watt hours

(2) Power to charge battery $= \dfrac{19.4 \text{ WH}}{95.5 \text{ H}} =$ 0.203 Watts

(3) S/C loads including prime conv. losses $= \dfrac{18 \text{ watts}}{0.85} =$ 21.2 Watts

21.40 Watts

Excess power required due to array-battery interface (from Figure 23) = 5.7 Watts

Total power req. at array = 27.1 Watts

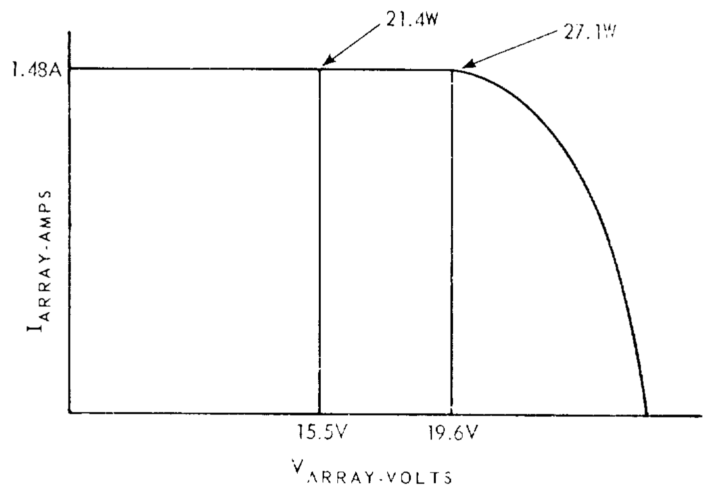

Figure 23. Battery-Array Loading Effects for the IMP Power System in a 4 Day Orbit

System Efficiency Calculation

$$\text{Con. Efficiency} = \frac{(85)\ (3.995)}{4} + \frac{(85)\ (75)\ (0.5 \times 10^{-2})}{4} = 84.8\%$$

$$\text{System Efficiency} = \frac{18}{27.1} = \boxed{66.4\%}$$

III. Array Maximum Power Tracking System

This system consists of a maximum array power tracking regulator, an 18-cell battery, and load regulator. The main bus regulator operates with an efficiency of 90% and provides a regulated 28 volts ±2% output. The regulator operates over a voltage range from 16.2 to 27 volts as defined by the charge and discharge characteristics of the battery. The maximum array power tracking regulator controls the array operating point so maximum array power is available when required.

A. MPT System 7.5 Hour Orbit

Condition A: Shadow (30 minutes)

		Record Mode	Playback Mode
(1)	S/C loads	18 watts for 18 min.	31 watts for 12 min.
(2)	Losses MBR (90% eff.)	2 watts	3.45 watts
(3)	Pwr. req. from batt.	20 watts	34.45 watts

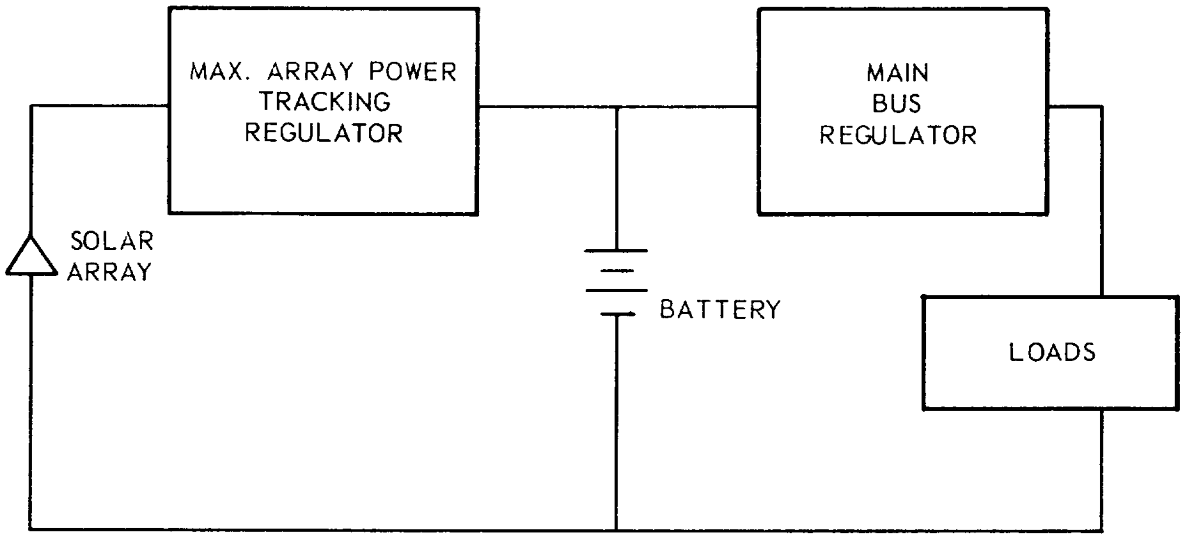

Figure 24. A Maximum Array Power Tracking Power System (MPT)

(4) Energy required from battery

$$20 \text{ W} \times \frac{18}{60} \text{ hr.} = 6.0 \text{ WH}$$

$$34.45 \text{ W} \times \frac{12}{60} \text{ hr.} = 6.9 \text{ WH}$$

Total 13.9 WH

Condition B: Sunlight (7 hours)

(1) Energy for charge $\frac{12.9}{(0.75)\ (0.9)} = 19.2$ WH

(2) Power req. for charge $\frac{19.2 \text{ WH}}{7 \text{ H}} = 2.75$ watts

(3) Load power required including APTR and MBR losses $\frac{18 \text{ watts}}{(0.9)\ (0.9)} = 22.2$ watts

24.95 $\approx$ 25 watts

(4) Array power required $= \frac{\text{Power req.}}{\text{Array utilization}} = \frac{25.00 \text{ watts}}{97\%} = 25.8$ watts

(5) Average power required at the array. Additional average array power gained from utilization of full power when array is cold = 0.4 watts for 7.5 hour orbit

$$25.8 - 0.4 = \underline{25.4 \text{ watts}}$$

System Efficiency Calculation

$$\text{Conv. Efficiency} = \frac{(E_{AR})\ (E_{LR})\ t_L}{t_T} + \frac{(E_{AR})\ (E_{LR})\ (E_B)\ t_D}{t_T}$$

$$\text{Conv. Efficiency} = \frac{(90)\ (90)\ (7)}{7.5} + \frac{(90)\ (90)\ (75)\ (0.5)}{7.5} = 79.6\%$$

$$\text{System Efficiency} = \frac{18.376 \text{ watts}}{25.4 \text{ watts}} = \boxed{72.4\%}$$

where:

E_{AR} - Array maximum power tracking regulator efficiency.

B. MPT System 90 Minute Orbit

Condition A: Shadow (30 minutes)

		Record Mode	Playback Mode
(1)	S/C loads	18 watts for 24 min.	31 watts for 6 min.
(2)	Losses BDR	2 watts	3.5 watts
(3)	Pwr. from Batt.	20 watts	34.5 watts

(4) Energy from battery

$$20 \text{ w.} \times \frac{24}{60} = 8 \text{ WH}$$

$$34.5 \text{ w.} \times \frac{6}{60} = 3.45 \text{ WH}$$

11.45 WH

Condition B: Sunlight (60 minutes)

(1) Energy required to charge battery $\dfrac{11.45 \text{ WH}}{.75 \times .90} = 17 \text{ WH}$

(2) Power to charge battery $= \dfrac{17 \text{ WH}}{1 \text{ H}} = 17.0$ watts

(3) Load power required including APTR and MBR losses $= 22.2$ watts

39.2 watts

(4) Array power required $= \frac{39.2}{97\%} = 40.5$ watts

(5) Average power required at array

Average power gained from utilization of full power when array is cold = 10%* of $\text{Power}_{\text{Max Power Point}}$ = 10% of 40.5 = 4.05

$$40.5 - 4.05 = 36.45 \text{ Watts}$$

System Efficiency Calculation

$$\text{Conv. Efficiency} = \frac{(90)\ (90)\ (60)}{90} + \frac{(90)\ (90)\ (75)\ (30)}{90} = \underline{74.2\%}$$

$$\text{System Efficiency} = \frac{18.87 \text{ Watts}}{36.45 \text{ Watts}} \quad \boxed{51.8\%}$$

C. MPT System 4 Day Orbit (32 Earth Radii)

Condition A: Shadow (30 minutes)

Energy required from battery = 13.9 watt hours

Condition B: Sunlight (95.5 hours)

(1) Energy to charge battery $\frac{13.9}{.75 \times .90} = 20.6$ WH

(2) Power to charge battery $= \frac{20.6 \text{ WH}}{95.5 \text{ H}} = 0.216$ Watts

(3) Load power required including APTR and MBR losses 22.2 Watts

22.416 Watts

(4) Array power req. $= \frac{22.416}{97\%} = 23.1$ watts

Power gained from utilization of full power when array is cold = 0.032 watts.

(5) Average array power required = 23.07 watts

*Based on the S^3-A thermal design. This percent is typical for the use of a body mounted array with good thermal transfer to the spacecraft structure.

System Efficiency Calculation

$$\text{Conv. Eff.} = \frac{(90)\ (90)\ (3.995)}{4} + \frac{(90)\ (90)\ (75)\ (0.5 \times 10^{-2})}{4} = \underline{81\%}$$

$$\text{System Efficiency} = \frac{18}{23.07} = \boxed{78.3\%}$$

IV. System Comparisons

Table 3 shows a comparison of the 3 systems with regard to system and conversion efficiencies for various orbit conditions. You will note from this table that in the orbits considered, the proposed system offers the best overall system efficiency, while the MPT system approaches the proposed system in

Table 3

Efficiency Comparisons of the Systems

7.5 Hour Orbit (30 min. shadow)			90 Min. Orbit (30 min. shadow)		4 Day Orbit (30 min. shadow)	
Sys.	Con. Eff. %	System Eff. %	Con. Eff. %	System Eff. %	Con. Eff. %	System Eff. %
S^3	97.5	87.6	86.8	54	99.9	98.9
IMP	83.4	60.6	77.9	39.9	84.8	66.4
MPT	79.6	72.4	74.2	51.8	81	78.3

system efficiency as the orbit time decreases and shadow time increases. Also you will find that the efficiency of the proposed system is quite high for near continuous sunlight orbits since no conversion stage is used between the array and the load. Figure 25 is a graph showing how the system efficiency varies as a function of orbit time for the 3 systems.

From the preceding calculations, a graph showing the array power required versus the orbital period was plotted in Figure 26. From this curve, it is seen that for the S^3-A orbit of 7.5 hours and spacecraft load of 18 watts, the proposed system requires an array power of 21 watts while the MPT system requires 25.5 watts and the IMP system 30 watts. To meet the S^3-A power requirements

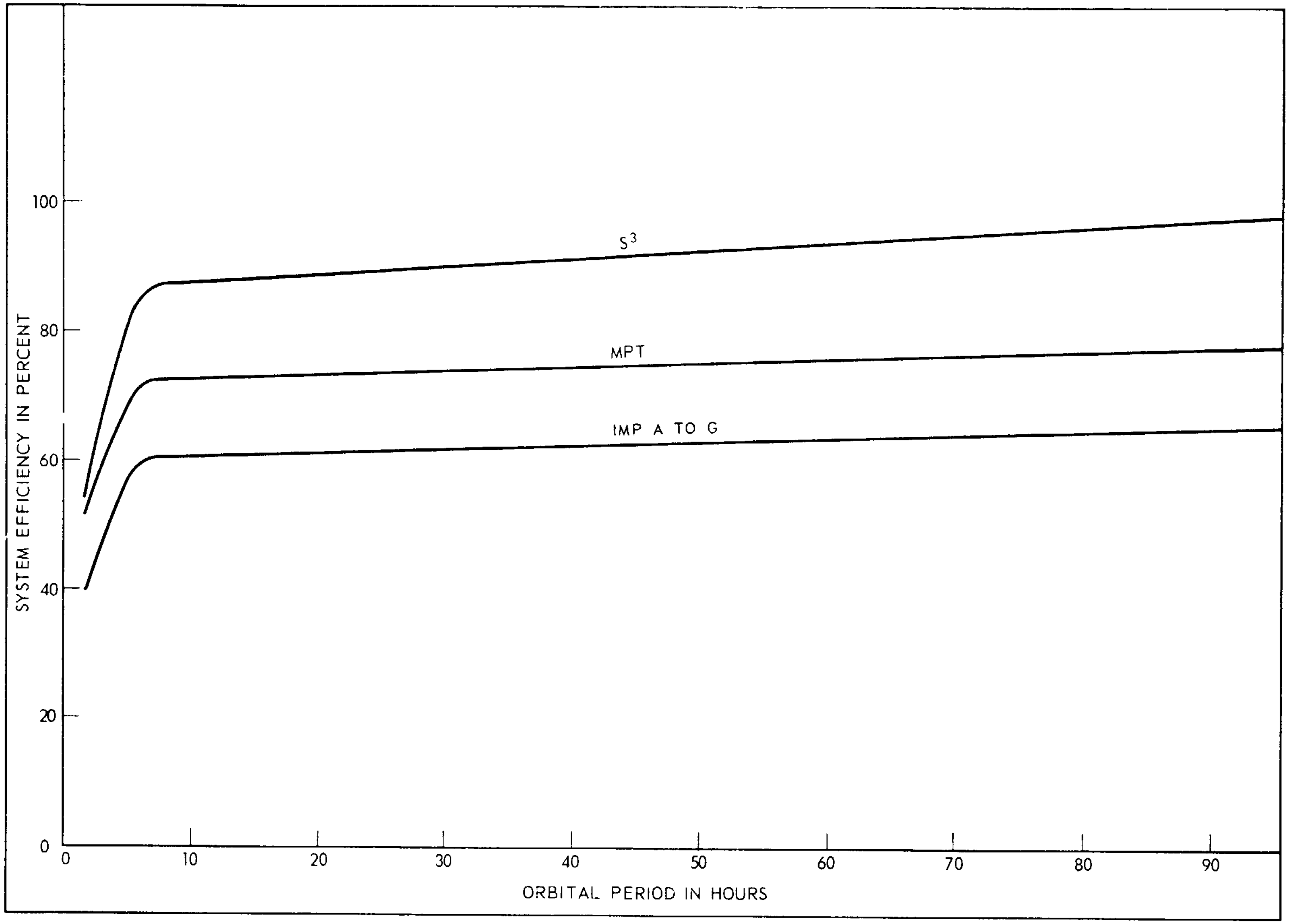

Figure 25. Graphical Comparison of the System Efficiency of the S^3, IMP, and MPT Power Systems as a Function of Orbital Period

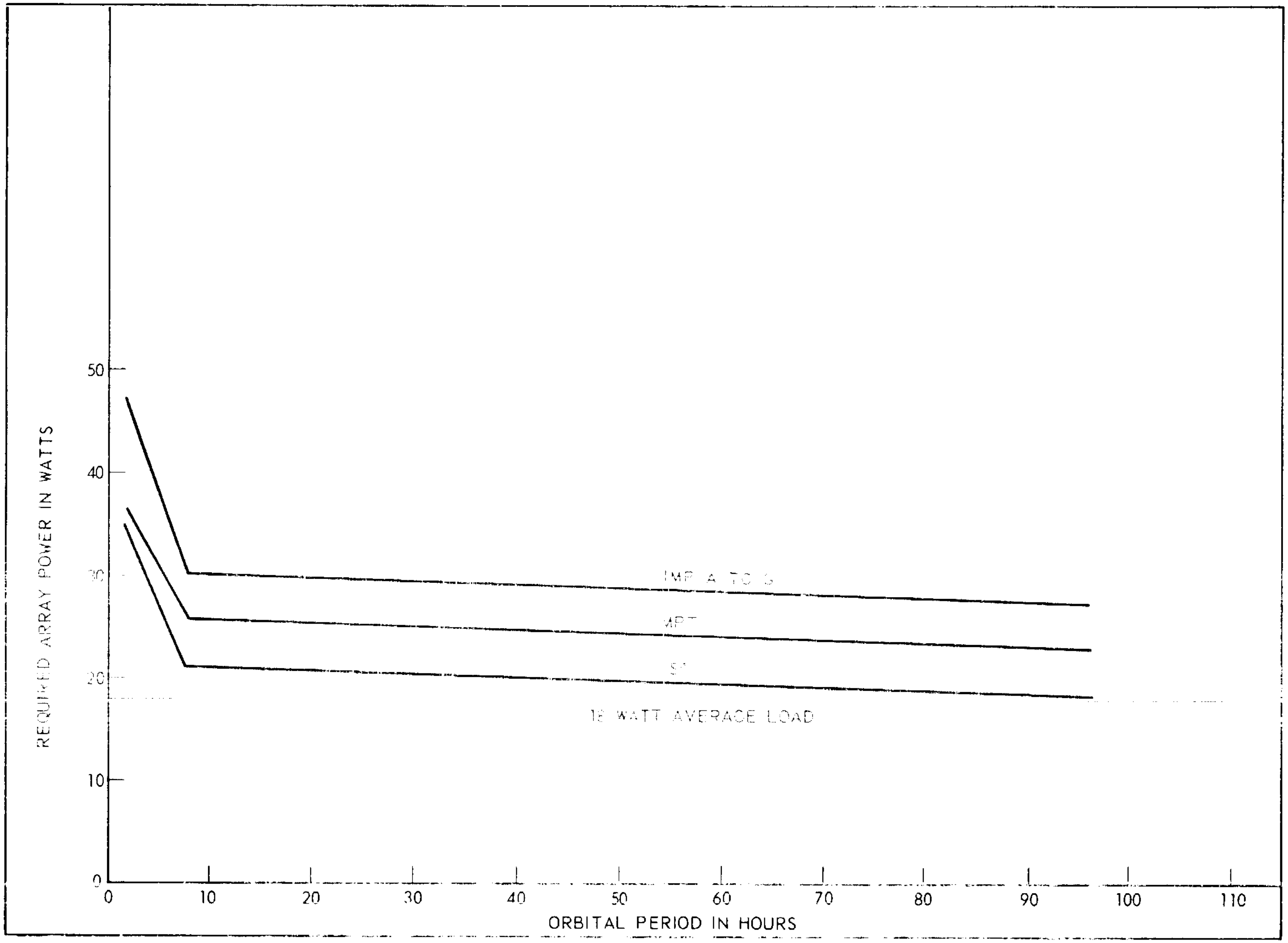

Figure 26. Graphical Comparison of the Array Power Required By the S^3, IMP, and MPT Power Systems As a Function of the Orbital Period

with the proposed system solar cells were mounted on all the available body surface. For a system requiring more solar power such as the IMP system, additional solar cell mounting area is required and could only be provided by adding paddles. This is undesirable since it results in increased weight and cost.

CONCLUSIONS

Detailed analysis of array-battery and load interfaces shows that maximum system efficiency is achieved when the battery is decoupled from the array. With this condition satisfied, the array can be operated at a fixed voltage point where the array is designed to provide maximum power. The main bus conversion step can be eliminated and the array used directly for load power. This provides a significant increase in efficiency, on the order of 8 to 10%.

Using this information, a highly efficient solar array battery power system has been designed for the S^3 spacecraft. This system uses a combination shunt-charge regulator and a battery discharge regulator to regulate the main bus to 28 volts ±2%.

In addition to high efficiency, the system provides the following advantages:

(1) Excellent utilization of array power which results in a significant reduction in the weight and cost of the array.

(2) Good array-battery static and dynamic load sharing when load power exceeds array capability.

(3) Flexibility to accommodate varying mission requirements such as low altitude orbit or power increases to 60 watts.

(4) Low electromagnetic interference during the sun portion since the series power switching regulator between the array and load is eliminated.

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